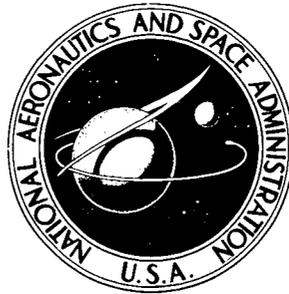


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APOLLO EXPERIENCE REPORT -  
LUNAR MODULE REACTION CONTROL SYSTEM

*by Chester A. Vaughan, Robert Villemarette,  
Witalij Karakulko, and Donald R. Blevins*

*Manned Spacecraft Center  
Houston, Texas 77058*

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16. Abstract <p>The design, development, and qualification of the reaction control system for the Apollo lunar module are described in this document. The lunar module reaction control system used many of the components developed and qualified for the service module reaction control system. The system was qualified for manned flight during the unmanned Apollo 5 mission on January 22 and 23, 1968, and has operated satisfactorily during all manned lunar module flights including Apollo 11, the first manned landing on the moon.</p>					
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# APOLLO EXPERIENCE REPORT

## LUNAR MODULE REACTION CONTROL SYSTEM

By Chester A. Vaughan, Robert Villemarette,  
Witalij Karakulko, and Donald R. Blevins  
Manned Spacecraft Center

### SUMMARY

The lunar module reaction control system was patterned very closely after the service module reaction control system. Components common to the service module reaction control system and the lunar module reaction control system were used where possible. Where components could not be common, common technology was used in the development of the lunar module reaction control system. The experience gained from Gemini missions and the command and service module reaction control systems in the areas of system fabrication, checkout, and testing also was applied to the lunar module reaction control system. The system reliability requirements were achieved through system and component redundancy. Two independent operational lunar module reaction control systems were provided.

The development and certification consisted of nine major ground test programs: (1) preproduction system development, (2) production system development, (3) design verification development, (4) production cluster environment, (5) lunar module production cluster firing, (6) integrated reaction control system/ascent propulsion system PA-1, (7) the NASA Manned Spacecraft Center in-house lunar module reaction control system test, (8) heater integration, and (9) engine valve temperature.

The checkout of the lunar module reaction control flight systems was divided into four basic categories of tests: (1) component, (2) module, (3) system level, and (4) vehicle integration checkout.

The performance of the lunar module reaction control system on Apollo missions was satisfactory. Several minor problems occurred, but solutions were found for all problems encountered.

### INTRODUCTION

The Apollo spacecraft is composed of the command module (CM), the service module (SM), and the lunar module (LM). In July 1961, the NASA Space Task Group released the first statement of work for the CM and the SM. Included in this statement

of work was a description of the reaction control system (RCS). The experience gained during the initial development of the CM RCS and the SM RCS was applied in the development of the LM RCS; and, as a result, the qualification program was simplified greatly. This technical note is concerned only with the development and evolution of the LM RCS.

## REQUIREMENTS

The Apollo missions required that the LM (fig. 1) maintain various attitudes with respect to its flight path and maneuver in three axes to achieve a successful manned lunar landing and return to the command and service module (CSM). Specifically, the LM was required to be stable during all phases of flight and to have three axes of translation available for CSM separation, for CSM docking, and for various translational maneuvers during the lunar-orbit rendezvous. In addition, X-axis longitudinal translation was required to provide propellant-settling thrust for the descent and ascent propulsion systems.

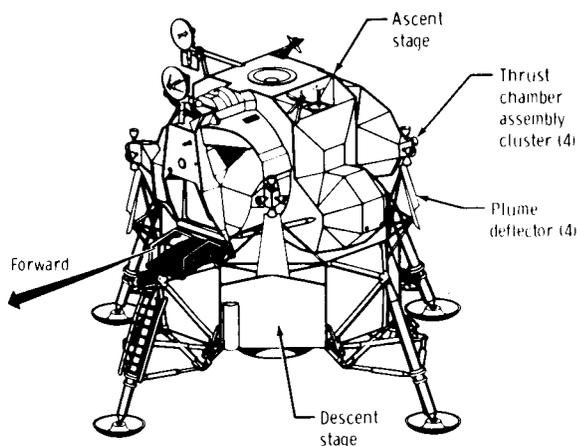


Figure 1. - Apollo lunar module.

A wide spectrum of operational requirements for vehicles that varied in mass and moment of inertia by a factor of 10, coupled with the necessary vehicle accelerations, established a requirement for rocket engines capable of producing high sustained thrust as well as low impulse. The initial design criteria were to provide rocket engines of various thrust levels to satisfy the variety of requirements. However, a close examination of the rocket engine that was being developed for the SM RCS revealed that the high thrust and low total-impulse capabilities could satisfy all the translational and rotational requirements of the LM mission. After this engine was selected, the remainder of the system was patterned very closely after the SM RCS; the differences were primarily in the propellant-

load capability and system geometry as dictated by the requirements of the LM mission. The similarity was extended to the use of common components wherever possible. Because of the common-use philosophy and the common-technology approach, component development and qualification testing for the LM RCS were simplified greatly.

The environmental constraints for the LM RCS generally were less severe than those of the SM RCS; therefore, the experience gained with the SM components in the areas of vibration, shock, thermal vacuum, compatibility with propellant, and susceptibility to contamination could be applied directly to the LM design. Two specific areas in which environmental conditions differed significantly were the vibration and the cold soaking of the four LM engine clusters. Also, because LM propellant tanks were considerably longer than the SM tanks and the helium tanks were larger in diameter, the vibration test experience with the SM tanks could not be applied directly to the LM hardware. In these instances, the components were subjected to environmental testing dictated specifically by the LM environments.

## DESIGN PHILOSOPHY

To ensure reliable system performance, the design of the LM RCS was based on system and component redundancy — similar to the Mercury and Gemini spacecraft and to the Apollo CSM. Two independent and operationally identical LM RCS systems, each capable of providing attitude control and positive and negative longitudinal translation, were provided (fig. 2). The RCS propellant supply (nitrogen tetroxide oxidizer and Aerozine-50 fuel) consisted of predetermined quantities for the lunar descent and ascent maneuvers. The tankage of each system was sized to contain one-half the RCS propellant required for descent, plus the total RCS propellant required for ascent. In addition, a contingency propellant supply was provided through an interconnect arrangement between the ascent propulsion system (APS) propellant tanks and the RCS manifolds. The interconnect arrangement originally was meant to be used only in an emergency situation. However, the interconnect arrangement was used as a normal operating mode during the powered-ascent phase to conserve RCS propellants for docking contingencies.

Within each pressurization system, redundancy was used for such components as regulators, check valves, and explosive pressurization valves. The explosive valves were in a parallel configuration because the primary failure mode was in a closed position. The regulators were in a series arrangement because the primary failure mode was in an open position. The check valves were arranged in a series-parallel configuration (1) because the failure probability in an open or closed position was considered to be about equal and (2) because the weight penalty associated with this particular component was minimal. Thus, no single functional failure could impair the control of the spacecraft or jeopardize crew safety because of propellant shortage.

The common-use philosophy was applied throughout the system design, using developed CSM components wherever possible. Whenever the component could not be used directly, but could be made usable on the LM with minor modification, a common-technology approach was followed. The manufacturer of the SM part was given the task of modifying his product to make it usable on the LM. Because this approach permitted the use of the same test procedures, test equipment, and personnel employed for the SM, the resources and learning time required to produce a given piece of hardware were minimized and, thus, significant cost savings and increased confidence in reliability resulted.

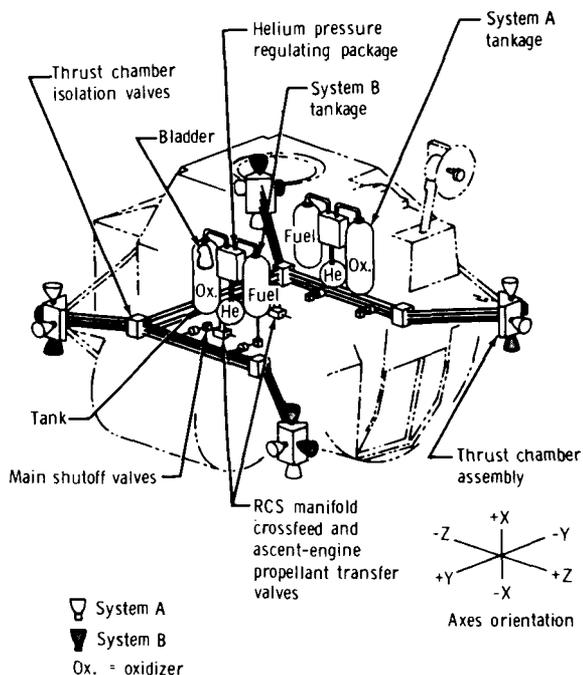


Figure 2. - Reaction control system installation.

# DESIGN

The LM RCS consisted of two independent systems, A and B. Each system provided the vehicle with attitude control and X-axis translation when used independently. When used together, Z- and Y-axis translation could be obtained also. The two systems were identical in all respects other than the engine locations and thrust vectors.

Each system had an independent helium-pressurization module, propellant tanks, and propellant manifold (fig. 3). The helium-pressurization module consisted of the helium storage tank, two parallel initiating explosive valves, a filter that protected the regulator, an orifice that acted as a damping device, a pressure regulator that reduced the initial storage pressure of 3000 psia to an operating level of 180 psia, a check valve assembly for each propellant (oxidizer and fuel) tank, and a relief valve assembly for each tank. The function of the check valves was to prevent backflow of fuel and oxidizer vapors, which later could condense in a common area and react to cause a local pressure and temperature rise. The function of the relief valves was to protect the propellant tanks from overpressurization in case of an increase in propellant temperature or a regulator malfunction. Each system included a number of servicing and checkout test ports.

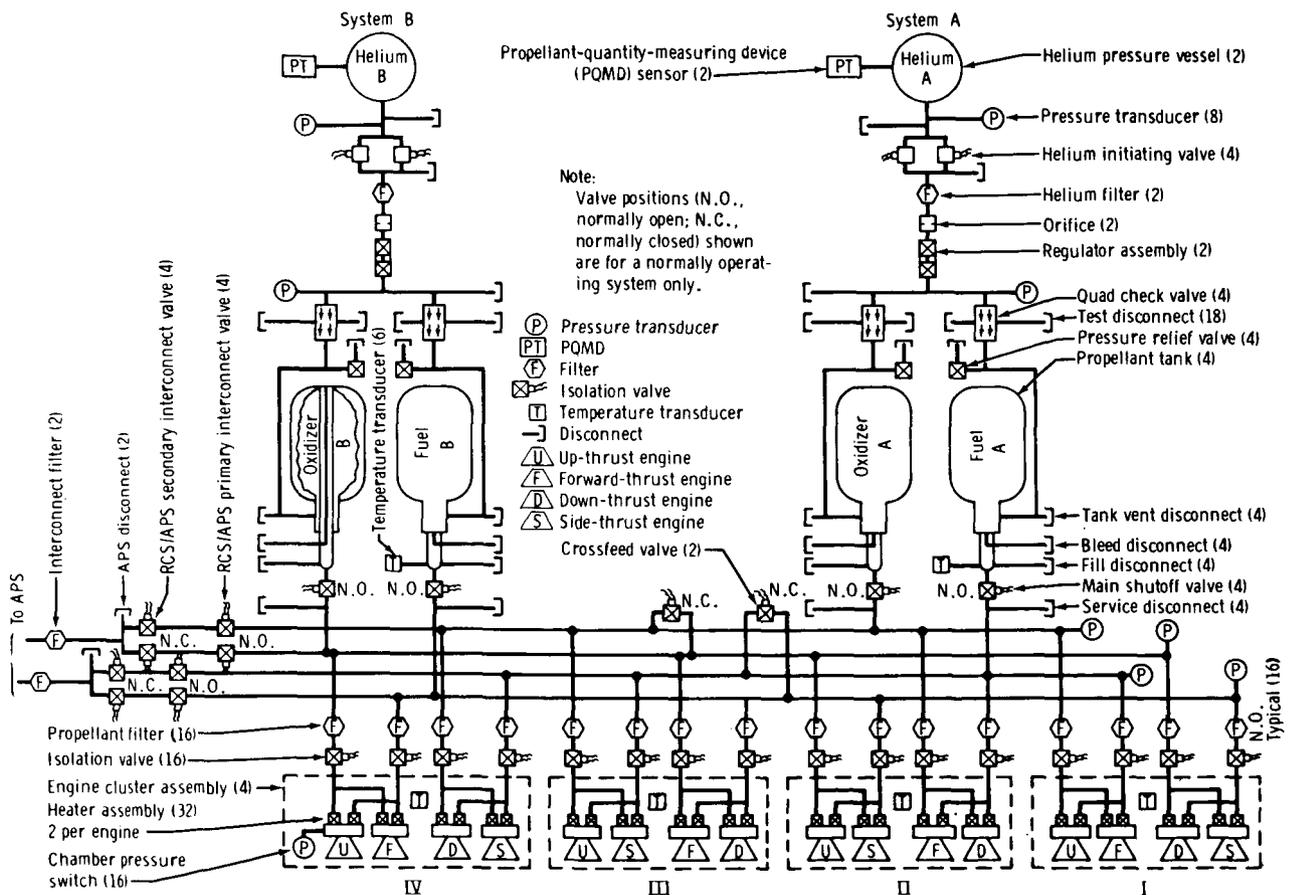


Figure 3. - Reaction control system schematic.

The propellant tanks were of the positive-expulsion configuration. Each propellant was contained inside a Teflon bladder that was in turn placed inside a titanium shell. Helium gas entered the area between the titanium and the Teflon, forcing the propellant out of the bladder and into a perforated standpipe connected to the tank outlet port.

The propellant manifold and distribution system is shown schematically in figure 4. The position of the main, interconnect, fuel, and oxidizer valves controlled the propellant distribution to the engines. The valves were developed specifically for the LM RCS applications because no component was available that could satisfy the requirements of every location. Each valve was a latching solenoid type, which required power only for opening and closing. After the desired position was established, the poppet was held in position by permanent magnets. The valves were well suited for nominal, static system conditions but were underdesigned for dynamic conditions; therefore, the crewmen had to verify correct valve positions during critical phases of flight. With proper crew procedures, the valves could be used for any dynamic environment without creating a major system problem.

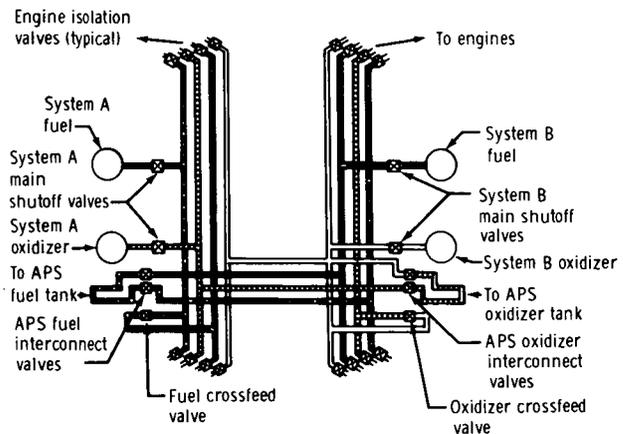


Figure 4. - Propellant manifold and distribution system schematic.

In-line propellant filters originally were located downstream of the isolation valves. However, because the isolation valves were particularly sensitive to contamination (as discussed in detail later), the filters were relocated upstream as shown in figure 3.

A redundant set of interconnect valves was provided. The valves ensured that, upon completion of the APS-interconnect operation during lunar ascent, the RCS propellants would not be transferred to the empty APS tanks and that pressurant gas from the APS tanks would not be ingested into the RCS. These interconnect valves were normally in the open position, and they were closed only if a malfunction was detected during the nominal APS-interconnect termination procedure.

## Flange Heater and Pressure Switch

Two components developed specifically for use with the LM RCS engines were the engine flange heater and the engine chamber pressure switch.

Engine flange heaters. - The engine flange heaters were required to maintain the temperature of the engine combustion-chamber flange above 120° F. This temperature level, which was determined during the engine-requalification program described later, was required to ensure safe operation of the engines with the Aerozine-50 fuel during all phases of the mission. Two heaters per engine were provided to ensure redundancy.

Both heaters were operated by the automatic system, and one heater had a manual on/off override.

Engine chamber pressure switch. - The engine pressure switch was used in conjunction with the failure-detection system of the LM. The electrical signal from the guidance, navigation, and control system to the engine valves was compared electronically with the output of the pressure switch; and, if the two did not match, the engine failure indication was displayed to the crew. Corrective action such as engine isolation or troubleshooting in other systems then could be performed by the crew.

### Installation of the System

The modular grouping of the propellant and helium storage components was made for the purpose of simplifying the checkout and repair procedures. Both operations could be performed "on the bench" without interfering with the overall vehicle operations. The system was installed in two bay areas and on four outrigger booms. The tankage modules (helium, fuel, and oxidizer) were installed on the left- and right-hand sides of the LM directly above the APS tanks. The engines were installed in clusters of four on the outriggers, which were located around the periphery of the ascent stage at 45° to the orthogonal (pitch and roll) axes. Two of the four engines in each cluster were fed from each propellant supply.

### DEVELOPMENT AND CERTIFICATION

The development and certification of the LM RCS consisted of nine major ground test programs. An overall RCS development schedule that includes the time phasing of the various programs and of the first six flights is shown in figure 5. A brief discussion of each major test program follows.

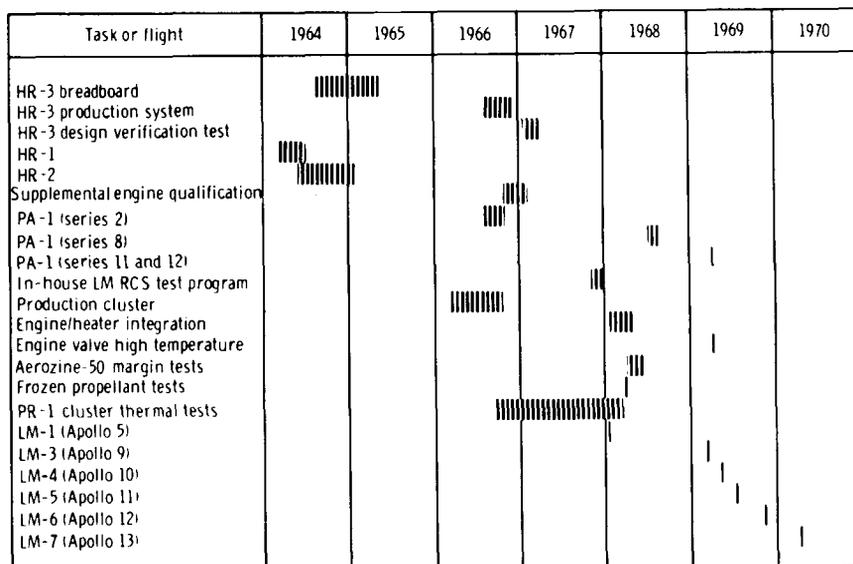


Figure 5. - Development and flight schedule.

## Preproduction System Development Test Program

The preproduction system development test program, or breadboard test, was the first test in which the proposed configuration of the LM RCS was hot fired. The term "configuration" is emphasized here because the geometrical configuration of the propellant-feed plumbing, which was the most important test item, was about the only aspect of the test hardware that resembled actual flight hardware. With the exception of the very early prototype engines, the breadboard system was composed entirely of commercially available industrial-type components.

In early 1964, when the breadboard was being assembled, most flight components were still in early stages of development and were rather scarce. Most of the components were being developed under a common-usage agreement, and those few parts made available to the LM program were being installed on the LM-1 vehicle and on major test vehicles.

The breadboard testing was conducted by the RCS engine developer between August 1964 and May 1965. One of the primary objectives of the test program was to investigate the dynamic characteristics of the propellant-feed system. The maximum pulse-frequency requirement for the engines was 25 pulses/sec at a pulse duration of 10 milliseconds. This high pulse frequency caused concern that the response of the relatively long propellant feedlines of the LM system was too slow to maintain adequate pressures at the engine inlets during transient flow. Other objectives were to evaluate propellant-manifold priming procedures and engine performance during multiengine firings. Another important aspect of the program was to "shake down" the new LM test facility and data-acquisition equipment, which had been installed specifically for LM RCS development testing.

The firing program consisted of single and multiengine firing matrices covering a wide range of pulse widths, pulse frequencies, and predicted flight duty cycles. The effects of pulse firings on the steady-state performance of another engine in the same system also were investigated. Almost 15 000 separate firings totaling approximately 8000 seconds of burn time were accumulated on the eight-engine system.

Although the breadboard configuration hardly resembled a flight system, it provided valuable data much earlier than would have been possible if the test program had been dependent on the availability of flight-type hardware. The breadboard test program was completed almost a year before the first flight-type configuration was available for testing.

One of the most significant findings of the test program was that the feed-pressure fluctuations during the short-pulse high-frequency firing were more severe than had been predicted analytically. In certain pulse modes, the transient engine-inlet pressures dropped to levels as low as the propellant vapor pressure, in which case no thrust was produced by the engine. On the other hand, peak transient pressure generated at engine valve closure approached the proof-pressure requirements of some components.

Fortunately, the detrimental effects of these severe feed-pressure fluctuations were recognized in time to avoid serious effect to the LM program. This information led to a complete reevaluation of the control system requirements for the LM and

helped to define the interface between the guidance system and the RCS. The significant result of the reevaluation was merely a change of the maximum pulse frequency from 25 to 7 pulses/sec.

Another significant conclusion from the breadboard testing was that the planned technique for filling or priming the propellant manifolds resulted in excessive transient pressures, in some cases greater than the design burst pressure for some flight components. The burst-pressure requirement for most propellant-feed system components was 550 psia. During priming, with nominal operating tank pressures, transient pressures of almost 1100 psi were recorded. These data resulted in modification of the planned flight-activation procedure from priming with full system operating pressure to priming at tank pad pressure, before activating the helium-pressurization system.

## Production System Development Test Program

The second major test program, the production system development test program, was conducted from August to November 1966, during the development of the LM RCS. This program was perhaps the most significant in the LM RCS development.

The test rig was designated HR-3P. With the exception of additional instrumentation and minor modifications necessary to facilitate ground test operations, the configuration of the HR-3P test rig was almost identical to that of the RCS on LM-1, the first flight vehicle.

The basic objective of the test program was to determine if the system could achieve fundamental design requirements. Most of the components in the test rig were prequalified or qualified models that had undergone extensive development tests as individual components. However, HR-3P was the first rig that was assembled and tested as a complete system.

To facilitate efficient test operations, the program was divided into a series of short tests, each with a specific objective relating to the various environmental or operational conditions that a flight system might experience during a lunar landing. Also, anticipated flight conditions were simulated whenever possible within the operational limitations of the test facility and the imposed schedule requirements. All tests were performed at local barometric pressure, which was approximately 12.5 psia. A large number of specific test objectives included the investigation of the following items.

1. Helium-pressurization-system activation and propellant section priming
2. System performance during simulated vehicle control modes and mission duty cycles
3. The effects of high and low temperatures on the performance of the system
4. System performance during crossfeed and simulated interconnect operation
5. Effectiveness of system-malfunction procedures and component redundancy

## 6. Component compatibility with propellants

## 7. Decontamination techniques and fluids

The HR-3P test rig also provided valuable experience in helium and propellant servicing. This experience was used in the design of the spacecraft ground-support equipment used at the launch site.

The firing program consisted of single and multiengine firings under nominal and off-nominal conditions of propellant-feed pressures, engine valve voltages, and propellant temperatures. Approximately 57 000 separate engine firings for a total of approximately 8300 seconds of burn time were accumulated on the system engines during the program.

Except for the rather routine operational problems with instrumentation and support facilities that are usually expected during a complex and large-scale test operation, the program ran rather smoothly and was completed within a reasonable period of time. All test objectives were accomplished; that is, the capability of the system design to meet fundamental requirements was demonstrated and no insurmountable deficiencies were uncovered. The unique environment resulting from system operation did disclose salient characteristics of some components that were not compatible with all planned system operational modes. An outstanding example was the discovery that the propellant latching valves would unlatch and shift position when subjected to the high flow rates or pressure surges that occurred during initial filling or priming of the propellant manifolds. This problem led to a very comprehensive investigation into the design characteristics of the latching valve and revealed the true limitations of the valve. As mentioned previously, this valve problem was solved for flight by requiring the crew to ascertain correct valve positions during critical phases of flight. Other component problems discovered were (1) transducer diaphragm incompatibility with propellant combustion residuals and (2) an inadequate seal design in the ground half of the propellant-servicing quick-disconnect couplings.

The test program also substantiated the contention that the contamination control requirements and procedures for the system were incompatible with the designs and reliability requirements of some components. Almost every system component and many facility components experienced leakage failures caused by particulate contamination. The propellant latching valve, because of a very narrow (0.006 inch) seat, was found to be particularly sensitive to contamination. The large number of leakage failures caused by particulate contamination provided support for a broad-based contamination control program, which is discussed in the section of this report entitled "Lunar Module RCS Checkout of Flight Vehicles."

## Design Verification Development Test Program

The third system-level test program was a design verification test (DVT) executed during February and March 1967. Whereas the objective of the previous production system test program was to determine the performance of the system under various conditions, the objective of the DVT program was considerably broader in scope. Not only was acceptable operation of the system demonstrated, but other factors such as

manufacturing and checkout procedures, contamination control techniques, and propellant decontamination procedures used on flight systems were verified. All components in the DVT system were fully qualified models and were assembled into the same configuration as that of LM-3, the first manned vehicle. Also, the DVT system underwent the same manufacturing and checkout operations as the flight systems.

The DVT program was essentially a repetition of some parts of the production system test program. High and low propellant-temperature tests, crossfeed operation, and simulated failure-mode operation tests were conducted. Approximately 23 200 engine firings were made, totaling approximately 3800 seconds of engine-burn time.

### Production Cluster Environmental Test Program

The production cluster environmental test program demonstrated the structural integrity of the engine cluster and vehicle mounting hardware. Extensive shock and vibration tests in all major axes were conducted on a complete production flight-type engine cluster and boom assembly.

With the exception of the failure of a chamber pressure transducer bracket, the cluster design withstood all the mission-level random and sinusoidal vibration loads to which it was subjected. Overstress vibration levels of up to 200 percent of specification requirements also were imposed on the cluster without causing any significant structural failures. Failure of the transducer bracket resulted in a bracket redesign that was retrofitted on LM-1, the first flight vehicle.

### Lunar Module Production Cluster Firing Test Program

The firings of a complete, flightworthy LM engine cluster (four engines) under simulated altitude conditions took place during the production cluster firing test program that was conducted in April 1966. The primary objectives of the program were to evaluate engine performance under more realistic flight conditions (particularly low ambient pressure) and to determine the heat-transfer characteristics of the cluster during steady-state and pulse-mode duty cycles. The thermal data were to be integrated into the cluster thermal tests that were being conducted in a thermal vacuum facility that simulated the space environment.

The firing program consisted of single and multiengine firings that simulated selected portions of expected mission duty cycles. In all, 1807 seconds of burn time and approximately 8500 firings were accumulated on the four engines in the cluster.

On April 27, 1966, during the low-temperature mission duty cycle part of the program, the combustion chamber of the upfiring engine was destroyed by an explosion that occurred during the start of a pulse. This failure resulted in the immediate termination of the program and the initiation of an extensive failure-analysis effort.

The analysis revealed that the failure was caused by a combination of conditions rather than by one single cause. The upfiring attitude of the engine, low engine temperatures, helium saturation of the propellants, short-pulse firings, and relatively

high test-cell ambient pressure were some factors found to contribute to the accumulation of nitrate compounds, which could cause high ignition overpressures and injector-manifold explosions.

The engine failure brought about an extensive engine-requalification program but did not result in any engine design changes. For the CSM application, the engine was qualified with monomethylhydrazine fuel. For the LM application, the requalification was done with Aerozine-50 because of the requirement for use of the same fuel for the LM RCS and APS. When the engine flange temperature was maintained above 120° F, engine failure was unlikely. The cluster heater design was changed to ensure 120° F flange temperature.

## Integrated RCS/APS PA-1 Test Program

The PA-1 test rig was a flight-weight ascent-stage structure with only the RCS and APS installed. It was built principally for development testing of the APS, and the RCS was included primarily for evaluating the interconnect-propellant-feed mode.

The configuration of the PA-1 RCS was similar to that of test rig HR-3P and flight vehicle LM-1 except that solenoid valves were installed between the propellant tanks and the helium module to prevent migration of propellant vapors into the regulators and check valves during extended downtimes. Also, additional feedline access ports were installed between the cluster isolation valves and the engines to aid in draining propellant.

All testing was conducted at the NASA White Sands Test Facility (WSTF) during September and October 1966. A series of 11 runs was made under the altitude conditions (88 000 to 140 000 feet) attainable with the WSTF vacuum pumping system. Approximately 3000 firings and 515 seconds of firing time were accumulated on 12 engines; the four upfiring engines were disabled to prevent possible injector-manifold explosions, which could occur at the relatively high test-cell pressure. The high test-cell pressure was not a realistic simulation of space vacuum.

Analysis of the data indicated that neither the RCS nor the APS experienced any detrimental effects during the interconnect-feed operation. A very minor decrease in RCS engine thrust, estimated to be 1 to 2 percent, was observed while propellant was being supplied from the APS. This condition was attributed to slightly lower engine-inlet pressures resulting from increased pressure loss through the longer feedlines. The propellant pressure transients generated during RCS engine pulsing were found to have little or no influence on the performance of the APS engine.

As in earlier system test programs, the lack of adequate contamination control was the only system-oriented problem that arose during the testing. Almost all the propellant latching valves in the system experienced internal leakage; and, before the testing began, all engines had to be returned to the manufacturer for cleaning.

A demonstration test of a proposed flight procedure for venting the RCS propellant manifolds of the nitrogen pad pressure by opening the engine valves at high altitude was unsuccessful. Gross propellant leakage through either the interconnect valves or the main shutoff valves resulted in hot firings during the attempted manifold venting. As a

result of this test and numerous leakage failures of the latching valve, the LM-1 manifold venting operation was eliminated.

The RCS testing on the PA-1 configuration was resumed with series 8, 11, and 12 tests, starting in July 1968 and ending in May 1969. The major emphasis during PA-1 series 8 testing was to test the APS and RCS in support of the first manned flight (LM-3). The RCS engines were operated in normal-feed, crossfeed, and interconnected-feed modes. The systems installed in the PA-1 rig were essentially the LM-3 configuration. All tests were conducted at simulated altitude conditions. The test-cell pressure was maintained below the 0.2-psia red-line value based on engine test experience; therefore, the upfiring engines could be fired. Testing consisted of base-line engine performance tests with various propellant-feed modes, high-altitude start testing, and selected firing matrices designed to evaluate the integration of the RCS and the caution and warning electronics assembly (CWEA).

The primary conclusions from series 8 tests were as follows.

1. The RCS performance in the various feed modes was acceptable.
2. Safe RCS start capability and acceptable engine performance at high altitude (220 000 feet) were demonstrated.
3. The NASA John F. Kennedy Space Center servicing procedures should be modified to include a continuous powering (open) of the main shutoff valves during priming of the RCS manifolds. All other priming and pressurization procedures were found to be acceptable.
4. Hydraulic interactions between the APS and RCS during interconnect feed caused minor fluctuations in the ascent-engine chamber pressure.
5. The integrated RCS/CWEA performed satisfactorily.

In series 11, RCS engines were operated in the interconnect-feed mode (APS propellants) throughout the series to simulate the ascent portion of the lunar mission. Testing consisted of a shakedown firing, an off-nominal lunar-landing-mission duty cycle to evaluate the extent of RCS-induced fluctuations on the ascent-engine chamber pressure, and subsequent heat-soakback tests to simulate various failed RCS engine configurations.

Series 11 testing indicated a potential problem with pressure rise of trapped propellant in the inlet manifolds as a result of thermal soakback from a hot engine. In these tests, thruster-pair isolation valves were closed and the RCS heater was turned off after RCS firing activity. In several cases, engine heat soakback and the concomitant thermal expansion of the trapped propellants resulted in inlet pressures up to the maximum allowed (700 psia in this test). The rate of pressure buildup showed that 700 psia would have been exceeded had the thruster-pair isolation valves not been opened for relief; consequently, the Apollo malfunction procedures incorporated pressure relief steps — firing of one of the two isolated engines after isolation of an engine pair.

Series 12 testing consisted basically of a shakedown firing, a simulated ullage burn (in support of the LM-3 APS anomaly investigations), an off-nominal lunar-landing-mission duty cycle (high-frequency RCS pulsing), and subsequent heat-soakback tests. It was concluded from series 12 testing that high-frequency pulsing (up to 11 pulses/sec) of the RCS does not initiate CWEA thrust chamber assembly (TCA) failure indications in either the normal- or interconnect-feed modes. Furthermore, high-frequency pulsing of the RCS does not seriously degrade engine performance, although the effect is greater in the interconnect mode than in the normal-feed modes.

## In-House LM RCS Test Program

A complete LM RCS test was conducted at the NASA Manned Spacecraft Center (MSC) in December 1967. The primary objectives of the test were to define the general operational characteristics of the LM RCS under simulated altitude conditions and to obtain performance data on individual subsystem components. This system test was the first to be conducted at simulated altitude conditions in excess of 100 000 feet.

The test article included all qualified components except the combustion-chamber pressure switches. Most system components and all propellant lines had been used previously in tests on the HR-3 DVT system at the subcontractor's Magic Mountain Test Facility. The HR-3 DVT configuration was modified and updated as required to satisfy specific test objectives and to incorporate the latest changes to the flight system. The most important of these changes were as follows.

1. Propellant-quantity-measuring devices were installed in each helium tank.
2. One flight-type thruster heater was installed on each engine.
3. A propellant filter was installed in each engine injector valve.
4. A pressure switch (not flight configuration) was installed in each of 16 engine injector heads.
5. Flight-type arc-suppression circuitry was installed on each engine.

The test program included (1) pretest operations, (2) base-line performance duty cycles, (3) simulated LM-1 and lunar-mission duty cycles, (4) special duty cycles designed to accomplish specific test objectives and evaluate system performance when subjected to "worst-case" duty cycles, and (5) post-test checkout and decontamination.

The primary objectives of the test were satisfied; that is, data on the general operational characteristics of the LM RCS and of the individual components were obtained. In the test program, three types of anomalies were observed. These anomalies, which were investigated and resolved, were propellant latch valve leakages, pressure switch failures, and engine injector cooling below the 120° F lower limit.

The propellant latch valve leakage was found to be caused by particulate contamination; therefore, the need for system cleanliness was emphasized. The switch failures were of two types — failed closed and failed open. The failed-closed condition was traced to the contamination of the switch mechanism by the semiliquid combustion

products. Such a failure would simply eliminate the usefulness of the switch but would not lead to a serious problem. The open failure was caused by a design deficiency that was corrected on the flight configuration. The injector cooling problem was traced to the engine duty cycle. The problem was resolved during later tests when it was shown that the normal mission duty cycles would not produce the cooling effect observed.

Another significant finding was that, in general, the HR-3 DVT system components performed within specification limits after testing at the subcontractor's plant and storage for several months at MSC. Also, following the test program, the system performance was adequate to complete a subsequent test program (LM-1 anomaly investigation) after a 4.5-month exposure to an unknown and uncontrolled concentration of propellants.

## Heater Integration Tests

After the engine was requalified with Aerozine-50 fuel at 120° F flange temperature, a complete cluster of four engines and eight engine flange heaters was subjected to numerous firing duty cycles. In the test program, certain combinations of short pulses caused the engine flange to cool faster than the heaters could warm it. In some cases, the cooling effect was so severe that the flange temperature dropped to below 100° F, and an engine explosion finally resulted. Additional test runs were performed to identify the flange cooling regimes. It was recognized that many of the duty cycles tested were much more severe than the duty cycle expected in flight. However, because the actual duty cycle in flight is highly unpredictable, it was necessary to establish the safe operating regime. The result was a map defining safe and unsafe engine operating duty cycles (fig. 6). Concurrent with the engine tests, additional mission simulations were performed to obtain a better estimate of the flight duty cycle. As can be seen in figure 6, the mission operating envelope (automatic mode) was found to be well within the safe region. The cooling effect was not a problem in the manual command mode because short-pulse combinations were not possible.

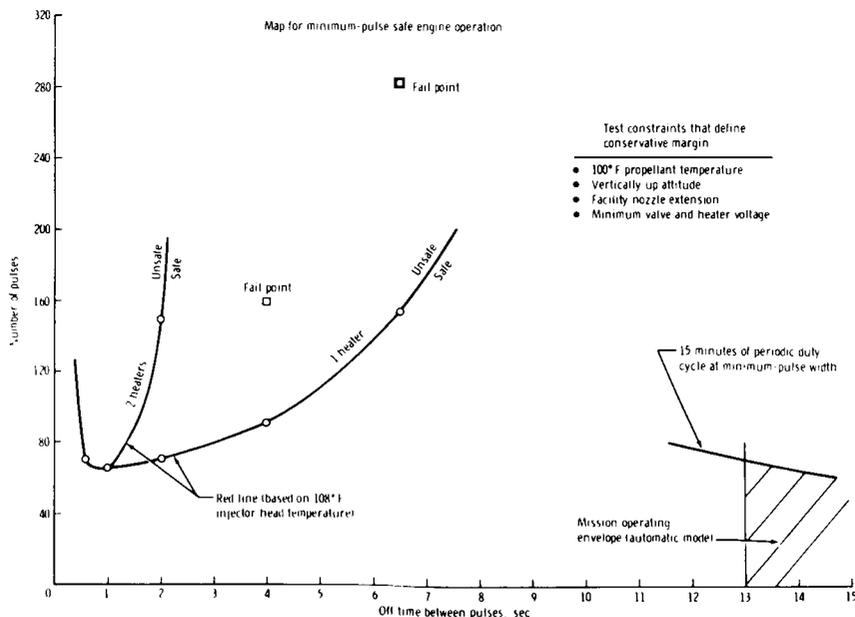


Figure 6. - Engine safe operating regimes.

## Engine Valve Temperature Tests

During the LM-3 mission, the temperatures recorded on the engine clusters exceeded not only the predicted values but also the upper limit of the cluster instrumentation. A series of tests was conducted in April 1969 to define a maximum temperature to which the engine valves could be subjected without degradation of performance. The engine was tested at ever-increasing temperatures starting at 275° F and ending at 375° F. The tests were terminated at 375° F when no degradation in performance was experienced. An instrumentation change (increase of upper limit to 260° from 200° F) also was made to accommodate the expected operating temperature of the clusters.

## LUNAR MODULE RCS CHECKOUT OF FLIGHT VEHICLES

The checkout of the LM RCS was divided into four basic categories of tests, which were conducted during the various stages of vehicle buildup. These were component tests, module tests, system-level tests, and vehicle-integration-checkout tests. In general, all tests conducted on individual components at each level of checkout were the same. Also, most test and success criteria were patterned after the component predelivery acceptance tests, which were derived from the original procurement specifications for the component. The various checkout tests performed in the course of the vehicle buildup provided increased confidence in the system and permitted the tracking of component performance from one test to another so that any degradation could be detected easily.

Component tests were conducted at two locations, at the plant of the component manufacturer and at the point of assembly just before installation into the vehicle. The preinstallation test (PIT) was essentially a repetition of the checkout by the component manufacturer with some of the less important tests eliminated. As the program progressed and more experience and confidence were gained with the hardware, the scope of the PIT was decreased gradually; and, in some cases, the test was eliminated entirely.

The PIT was informal and consisted of proof-pressure tests, external and internal leak tests, cleanliness verifications, and functional checks. Each test was conducted according to a test plan or test outline that did not call out each individual step of an operation, and these documents did not require the degree of quality control and inspection required by the higher level test procedures. This lack of formal test procedures was the most significant flaw in the PIT operation and resulted in the introduction of particulate matter into some components, rupturing of numerous helium-relief-valve burst disks, and shorting of position indicator switches on the propellant latching valve. However, the PIT did eliminate many faulty units before they could be installed on the vehicle.

The second level of checkout — the module level — was conducted on assemblies consisting of two or more components. Interconnecting brazed joints and plumbing were proof and leak tested, and the standard checks on individual components were repeated. The module-checkout concept increased the efficiency of the checkout operations by substantially reducing the complexity of the procedures and the test equipment.

For example, the elaborate and time-consuming precautions required to prevent flexure of the propellant tank bladders during regulator and relief valve checkout were completely eliminated in the helium-module checkout because this checkout was performed without the tankage module attached. The module-test concept also allowed flexibility in the checkout flow; that is, the various modules could be checked out independently without constraining other checkout functions. Other items checked as modules were cluster isolation-valve/filter assemblies and the propellant manifolds. Tankage-module checkout consisted of bladder leak checks, unlatch current measurement and leak tests of the main shutoff valves, and proof and leak tests of the brazed joints connecting the completely assembled and tested helium module to the tanks. Final instrumentation checks produced a complete tankage module ready for installation on the vehicle.

Concurrently with tankage-module checkout, the propellant manifolds were installed and verified clean by a liquid Freon flush. This cleanliness test was conducted in two stages. The first part verified the plumbing between the outlet of the tankage module to the inlet of the cluster filter, and the second part verified the cleanliness of the entire manifold up to the engine inlets. It should be pointed out that this liquid-flush cleanliness verification was incorporated only after a gaseous verification approach proved to be totally inadequate. The effectiveness of the flushing procedure was enhanced further by simultaneous low level vibration of the plumbing. Leakage failures of engine valves and propellant latch valves were reduced drastically by the liquid-flush techniques.

System-level tests consisted of manifold integrity tests, engine leak tests, and engine gas flow tests. Some helium-module components, which were susceptible to degradation by other functions or by time, also were retested. A rather unique test tool called a "calibrated feather" was used to verify, qualitatively, gas flow through each individual propellant orifice in the engine injector head. Until this device was developed, no method was available to determine if the small injector orifices were unobstructed.

Vehicle integration tests, usually conducted just before vehicle shipment, verified the interfaces between the RCS and the other systems. Engine valve wiring and response, cluster heater current draw, and instrumentation were checked during the final phase of factory checkout.

## LUNAR MODULE RCS FLIGHT PERFORMANCE

The LM RCS performance on all Apollo flights was satisfactory. Several minor problems occurred, but satisfactory solutions were found for all problems noted. The LM-1 and LM-3 flights indicated that the upper temperature limit of 190° F on the engine cluster was exceeded on numerous occasions with no deleterious effects. As previously discussed, additional vendor test data demonstrated that the engine valves could be safely operated at a much higher temperature. Consequently, the limit was deleted on LM-4 and subsequent vehicles.

Pressure switch "closed" failures occurred during the LM-3, LM-4, and LM-5 flights, but these failures had no significant effect on the flights because the only consequence was the loss of capability to detect an engine "off" failure. The single

pressure switch "open" failure that occurred intermittently on LM-5 was considered to be a serious problem because an erroneous TCA failure indication resulted. This problem was solved on subsequent missions by briefing the crews about the possibility of an erroneous TCA flag. The switches were kept in the system primarily to aid flight controllers in the real-time analysis of control problems and RCS engine operation during the critical powered-descent mission phase. They were considered to be the best switches available for this function but were not as reliable as the engine. A brief discussion of the LM RCS performance on the Apollo 5, 9, 10, and 11 missions is provided in the following sections.

## Lunar Module 1 (Apollo 5 Mission)

The unmanned Apollo 5 spacecraft, including the first flight-configuration LM, was tested successfully in earth orbit on January 22 and 23, 1968. The primary objectives of the Apollo 5 mission were to flight verify the LM ascent and descent propulsion systems and the abort staging function for manned flight. These objectives were accomplished. The LM-1 RCS configuration differed from subsequent LM RCS configurations in several areas of feed system design and instrumentation design.

During the mission, the RCS performance and operation were nominal until control of the spacecraft was switched to the guidance, navigation, and control system after abort staging (intentional separation of the ascent and descent stages of the LM). At that time, the vehicle mass in the digital autopilot was configured for control of a two-stage, fully loaded vehicle; consequently, the system was commanded to deliver propellant at a rate approximately 10 000 times greater than actually required. This anomaly caused the RCS to operate in several off-limit conditions and resulted in failures in the system. Within 3.1 minutes, the system A propellant was depleted to 27 percent, and that system was isolated to conserve propellant. System B continued at a rapid duty cycle until propellant depletion 5 minutes later, at which time helium started leaking through the collapsed system B fuel bladder. Satisfactory vehicle rates were restored by the system B thrust reduction (resulting from propellant depletion) and by the isolation of system A propellant tanks. While system B was operating with two-phase oxidizer and helium-ingested fuel, the quad 4 upfiring engine failed. When system A was reactivated, the system A main shutoff valve on the oxidizer side inadvertently closed. The ascent propellant interconnect valves were later opened, returning operation of the engines to normal until the interconnect valves were closed. The depletion of all propellant during the last minutes of the second ascent-engine firing allowed the spacecraft to tumble. Each of these specific RCS anomalies (i. e., the bladder, the engine, and the oxidizer main shutoff valve failures) was duplicated when a ground test system was exposed to similar duty cycle and environmental conditions after the flight.

The total propellant consumption from the RCS tanks was approximately 600 pounds. An additional 230 pounds of propellant were used from the APS tanks during interconnect operations. It is estimated that the RCS engines accumulated 16 000 firings during the mission. All inflight LM-1 RCS problems were considered to be a result of the operational anomaly that caused the RCS to operate in severe off-limit conditions. Therefore, no system design changes were made as a result of these problems.

## Lunar Module 3 (Apollo 9 Mission)

The Apollo 9 mission was the second to include the LM and the first to include a manned LM. The successful earth-orbital mission lasted approximately 241 hours from launch on March 3 to splashdown on March 13, 1969. The objectives of the mission were as follows.

1. Evaluate LM systems performance
2. Evaluate LM functional capability
3. Perform selected CSM/LM operations (rendezvous and docking)

These objectives were accomplished.

The LM RCS performed satisfactorily throughout the mission. The only problem noted was a failed-closed thrust chamber pressure (TCP) switch, which was used to monitor the quad 4 upfiring engine.

The total propellant consumption from the RCS tanks was approximately 353 pounds as measured by the onboard propellant-quantity-measuring devices. An additional 99 pounds were used from the APS tanks during interconnect operations. It is estimated that the RCS engines accumulated a total of 1250 seconds "on" time and 20 000 firings during the mission.

A significant decrease in the natural frequency of the LM RCS fuel and oxidizer manifold-pressure fluctuations was noted during interconnect-feed operations associated with the APS burn to depletion. The decrease apparently was caused by either free helium entering the RCS manifolds from the APS or a higher saturation level of APS propellant relative to RCS propellant. In any event, the condition was not detrimental to RCS operation.

## Lunar Module 4 (Apollo 10 Mission)

The Apollo 10 mission was the third to include the LM and the second to include a manned LM. The successful lunar-orbital mission lasted approximately 192 hours from launch on May 18 to splashdown on May 26, 1969. The objectives of the mission were to demonstrate the satisfactory performance during a manned lunar mission of the crew, the space vehicle, and mission support facilities and to evaluate the LM performance in the cislunar and lunar environments. These objectives were accomplished. The LM RCS performed satisfactorily throughout the mission. The only problem noted consisted of five failed-closed TCP switches.

The total propellant consumption from the RCS tanks was approximately 557 pounds; however, only 276 pounds of the total were used during manned operations. An additional 42 pounds of propellant were used from the APS tanks during interconnect operations. It is estimated that the RCS engines accumulated a total of 1640 seconds "on" time and 20 000 firings during the mission.

## Lunar Module 5 (Apollo 11 Mission)

The Apollo 11 mission was the fourth to include the LM and the third to include a manned LM. The successful lunar landing mission lasted approximately 195 hours from launch on July 16 to splashdown on July 24, 1969. The primary purpose of the mission was to perform a manned lunar landing and to return the crew safely to earth. This objective was accomplished.

Performance of the LM RCS was satisfactory throughout the mission. The only problem noted involved two TCP switches. One of them was a failed-closed or "on" condition similar to the experience on previous flights. The total propellant consumption from the RCS tanks was approximately 319 pounds. An additional 69 pounds of propellant were used from the APS tanks during interconnect-feed operations associated with lunar lift-off. It is estimated that the RCS engines accumulated a total of 1060 seconds "on" time and 12 000 firings during the mission.

During an 18-minute period just before terminal phase initiation, the quad 2 aft-firing engine switch failed to respond to seven consecutive minimum-impulse commands. This situation resulted in a master alarm and a TCA warning flag, which were reset quickly by the crew. Engine operation was nominal, and the switch failure had no effect on the mission. Subsequent crews were briefed that an erroneous TCA flag was possible and that, therefore, they should not abort unless the engine failure was verified by vehicle dynamics or some other means.

### CONCLUDING REMARKS

Successful completion of the ground-test program, coupled with excellent flight performance of the reaction control system, proved the system to be highly reliable. To a large extent, this high degree of reliability can be attributed to the commonality philosophy applied to the command and service module and lunar module reaction control system components. Another significant factor was the system and component cleanliness levels that were maintained by flushing and by providing in-line filters upstream of critical components.

The lunar module reaction control system did not experience significant design changes during development, qualification, or flight. This fact can be attributed primarily to application of the common-use and common-technology philosophy.

The engine injector valves proved to be extremely reliable. In the total program (both lunar module and service module), no engine injector valve leakage, as a result of engine operation or malfunction (explosion), was observed. The engine isolation valves, which were incorporated into the system to deal with the leaking engine valves, thus became of little value. It was concluded that the valves were not essential to crew safety, and the decision was made to delete them from the system in later flights to realize a 25-pound weight saving.

Manned Spacecraft Center

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